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DEPARTMENT OF MECHANICAL ENGINEERING AND MECHANICS
SCHOOL OF ENGINEERING
OLD DOMINION UNIVERSITY
NORFOLK, VIRGINIA

AN INVESTIGATION OF AERODYNAMIC CHARACTERISTICS OF WINGS HAVING VORTEX FLOW USING DIFFERENT NUMERICAL CODES

By

Sushil K. Chaturvedi, Principal Investigator

and

Farhad Ghaffari

Final Report
For the period ending March 31, 1984

Prepared for the
National Aeronautics and Space Administration
Langley Research Center
Hampton, Virginia 23665

Under
Research Grant NSG-1561
John E. Lamar, Technical Monitor
TAD-NTF Aerodynamics Branch

(NASA-CR-173596) AN INVESTIGATION OF AERODYNAMIC CHARACTERISTICS OF WINGS HAVING VORTEX FLOW USING DIFFERENT NUMERICAL CODES Final Report, period ending 31 Mar. 1984 (Old Dominion Univ., Norfolk, Va.) 30 p N84-25639 Unclas G3/02 00919

April 1984

OLD DOMINION UNIVERSITY RESEARCH FOUNDATION

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Submitted by the
Old Dominion University Research Foundation
P.O. Box 6369
Norfolk, Virginia 23508

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A	aspect ratio
b	span
C_r	root chord
C_t	tip chord
C_L	lift coefficient
ΔC_p	lifting pressure coefficient
D	drag force
L	lift force
M	Free stream Mach number
V_x	x-component of the total velocity vector
V_y	y-component of the total velocity vector
V_z	z-component of the total velocity vector
α	angle of attack
Λ	wing sweep angle

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Sushil K. Chaturvedi[1] and Farhad Ghaffari[2]

ABSTRACT

Three different numerical codes, namely, the free vortex sheet method, the quasi-vortex lattice method and the vortex lattice method with suction analogy, were employed to determine the aerodynamic characteristics of wings with separation induced vortex flows. Both flat as well as cambered wings of various planform shapes were studied. The effects of wing thickness, fuselage, notch ratio and multiple vortex modeling on aerodynamic performance of the wing were also examined. The theoretically predicted results were compared, wherever possible, with experimental results to validate the various computer codes used in this study.

A major task during the latter part of the grant period was to develop an analytical procedure for designing aerodynamically effective leading edge extension (LEE) for a thick delta wing. The LEE was located along the pseudo-stagnation stream surface (PSSS) corresponding to the design lift coefficient of 0.25 ($\alpha=6^\circ$) and free stream Mach number of 0.8. The PSSS was determined by employing the PAN AIR code developed by the Boeing Company.

The aerodynamic performance parameters of several LEEs with different

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constant chords, sweep angles and span extents were determined by excercising the vortex lattice method with suction analogy. Based on analytical results, the 0.8 constant chord LEE with 89% span extent was determined to be the best candidate for the selected wing.

INTRODUCTION

This report is organized into two parts. The first part presents the brief summary of the work performed during the grant period 1978-82*. During this phase, three different numerical codes, namely, the free-vortex sheet (FVS) method (refs. 1 and 2), the quasi-vortex lattice (QVL) method (refs. 3) and the vortex lattice method with suction analogy (VLM-SA) developed at NASA Langley (refs. 4 and 5), were employed to determine the aerodynamic characteristics of wings with separation-induced vortex flow. Special effort was made to validate these methods, and to establish their range of applicability with respect to aerodynamic parameters such as angle of attack, wing planform shape, thickness, camber, aspect ratio etc.

In the second part of this report, results obtained during the period Oct. 1982 - Mar. 1984** are summarized. Part of the work performed during this period was also funded by NASA Contract NAS-1-17099, Task Order 26, with Dr. J. Lamar of NTF aerodynamics branch as the technical monitor. The work accomplished during this phase is concerned with the design of an aerodynamically effective leading edge extension (LEE) for a thick, cambered, twisted wing of approximately triangular planform. The PAN AIR Code (ref. 6) was employed to determine the PSSS along which the LEE was located. The VLM-SA Code was used to determine the size and shape of an aerodynamically effective LEE for the chosen wing. In this report only the salient features of the LEE analytical design methodology are presented. The details of this subject matter are being reported in a companion technical report (ref.7). A complete list of publications during the grant period is given in Appendix A.

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**Principal Investigator: Dr. S. K. Chaturvedi

RESULTS AND DISCUSSION

Application of FVS, QVL and VLM-SA Codes

In this section, the results obtained from different numerical codes are summarized. The detailed discussion of results can be found in previous technical reports (refs. 8-11) submitted to NASA/Langley Research Center. A summary of wing configurations investigated in this study is given in tables 3-5 of ref. 8, tables 1-3 of ref. 9, table 1 of ref. 10 and table 1 of ref. 11. The angle of attack range over which a given code is employed, the type of method used, and whether convergence is achieved or not, are also indicated in the respective tables. It should be pointed out that initially the six parameter version of the free vortex sheet method, FVS(6-PV), was employed. Subsequently, an improved version employing nine parameter input, FVS(9-PV), was used as it became available. The results obtained from the FVS, QVL and VLM-SA Codes can be summarized as follows.

1. For flat wing planforms of low aspect ratios and at low to moderate angles of attack, all three codes predicted aerodynamic characteristics that are in good agreement with the data. However, pitching moment is generally overpredicted by all three codes.
2. The spanwise pressure distribution are generally not in good agreement with available data. For example, for a flat delta wing, the QVL method does not give the experimentally observed pronounced peak in ΔC_p. The FVS method gives a pronounced peak, but the peak values are overestimated compared to the data (refs. 8 and 9). The VLM-SA code gives only the integrated values rather than the local ΔC_p values.

3. The FVS method required comparatively large computational time and input data. The QVL method needed the least input data. The VLM-SA method requires the least amount of computational time, and it can handle complex planforms for which the other methods may not yield a converged solution.
4. Due to paucity of data for the configurations tested in the higher angle of attack range, no firm conclusions regarding the code capabilities in that range can be made.
5. The FVS(6-PV) method is unable to handle wings with large aspect ratios, low leading edge sweep angles and/or streamwise tips. The iterative scheme of the FVS (9-PV) method gives converged solution for the wings with streamwise edges after these edges are slightly modified and replaced by smooth curves. This task can also be accomplished by the least square scheme of the FVS (9-PV) method without even modifying the geometry. However, the least square scheme is more expensive to run and for this reason, the iterative scheme with slightly modified wing geometry is preferable.
6. The FVS(9-PV) method failed to yield converged solutions for most of the highly cambered configurations considered in this study (refs 8 and 9). In some cases, the selection of the starting solution affected the final solution. The QVL code was more successful in modeling highly cambered wings (ref. 9), although agreement between the theoretical results and the data was not always satisfactory. The VLM-SA code is capable of predicting results for highly cambered wings that are in good agreement with data for low to moderate angles of attack.
7. Using the multiple vortex modeling feature of the FVS(9-PV) method

for the flat double delta wing, better results compared to a single vortex modeling were obtained. However, the convergence was slower in the multiple vortex case. Converged results could not be obtained when the multiple vortex modeling feature was applied to a double arrow wing. It appears that the swept back trailing edge was responsible for the non-convergent behavior. The method also failed to give converged solution for a 74° flat delta wing with leading edge flap with two modeled vortices, one starting from the end of the leading-edge flap and another from the hinge line.

8. A series of flat wings with trailing edge notch were investigated using the FVS method. The effect of increasing notch ratio is to decrease the lift and pitching moment coefficients. The span loadings are unusual in comparison with the attached flow conditions in the sense that the slopes of the curves near the leading edge do not tend to infinity as they do in attached flow (ref. 9).

9. The QVL method can model low aspect ratio arrow, delta and diamond wings, but more study is required before it can conveniently be used for arrow and diamond wings. It is also interesting to note that this method could handle some of the high aspect ratio wings provided the angle of attack was also high. For example, the method could not handle an aspect ratio of 4 wings at an angle of attack of 10.36 or 20°, but it could handle the same wing at an angle of attack of 40°.

10. Converged solutions using the FVS(9-PV) code for a 74° flat delta wing with two inch conical leading edge vortex flap are also obtained. Several upward and downward flap deflections have been examined.

11. The effect of fuselage on the aerodynamic performance of SCAT-15F model (ref.10) wing-body combination was considered by using the FVS(9-PV) method. Source type networks were used to represent the body and the doublet-type networks for the wing. The effect of the body is to move the pressure peak locations outboard. However, the body effect on the longitudinal aerodynamic characteristics is not appreciable as indicated in table 2 of ref. 10. For the flat delta wing-body combination, the longitudinal aerodynamic characteristics as predicted by the free vortex sheet method did not compare favorably with the data (ref. 11).

12. The effect of wing thickness on the aerodynamic performance of SCAT-15F wing was also modeled by the FVS method. Though the converged solution was obtained, the predicted results were very much different from the data and also from the results obtained from the thin wing modeling. However, as mentioned in an earlier report (ref. 11), there was a programming error in the source panels used for modelling the wing thickness. Due to this uncertainty, no definite conclusion can be drawn regarding the effect of wing thickness.

13. The DM-1 glider (ref. 12) was also modeled by using the free vortex sheet method. It is a thick and round-edged wing of approximately 60° delta planform with NACA 0015-64 airfoil sections and aspect ratio of 1.8. Initially, the flat DM-1 with leading edge extension was modeled by the FVS method. As shown in fig. 1, the sum of squares of residuals (SSR, see ref. 1) was reduced from 10.0 to about 1.0 after four iterations using the least square solution technique. This solution was continued for four more iterations.

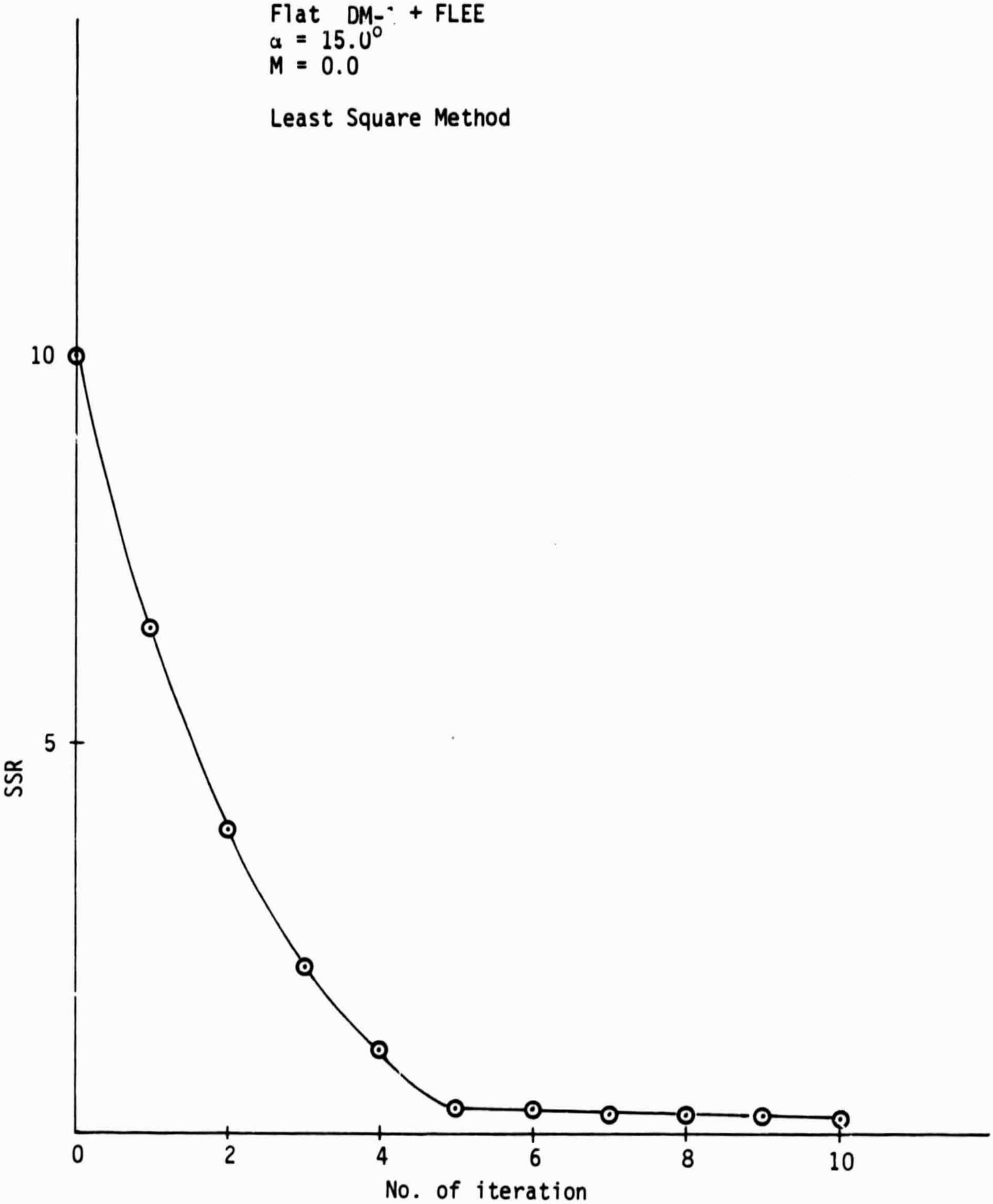

Figure 1. Variation of SSR with Number of Iterations

The additional iterations reduced the SSR to only 0.23 which is not normally considered to be a converged solution.* It was then decided to proceed with further iterations by adding the thickness to the basic DM-1 configuration. Two more iterations reduced the SSR to about 0.20 and the computer output indicated that the best possible solution has been achieved. The aerodynamic characteristics obtained for the thick DM-1 glider with leading edge extension are compared with data at $\alpha = 15°$ and $M = 0.0$ in table 1. The results apparently are not in good agreement with the data.

Table 1. Comparison of aerodynamic characteristics of thick DM-1 with leading edge extension at $\alpha = 15.0$ and $M = 0.0$.

Method	C_L	C_D	C_M
FVS	0.74	.20	.045
Data (ref. 12)	0.55	.12	-.038

*The solution is considered to be converged if the SSR value is of the order of 10^{-3} or less.

ANALYTICAL PROCEDURE FOR DESIGNING LEADING EDGE EXTENSIONS FOR THICK DELTA WING

An analytical procedure for designing leading edge extensions for a thick delta wing is developed. In order to demonstrate the analytical design methodology, a highly cambered and twisted wing of approximately triangular planform, 58° sweep angle and aspect ratio of 2.3 is chosen. The wing planform is shown in figure 2.

The LEE design is carried out in two steps. The first step involves determination of the pseudo-stagnation stream surface (PSSS) corresponding to the attached flow condition at lift coefficient of 0.25 for the selected wing. It should be remarked that the PSSS is the dividing stream surface which generally divides the oncoming flow over the upper and under the lower wing surface. The task of PSSS determination is accomplished by employing the PAN AIR code, and following the completion of this task, the LEE is located along the PSSS. If the PSSS is determined accurately, one would expect the LEE to have little effect on the wing aerodynamic performance at design condition, except for a slightly increased value of the drag coefficient due to skin friction of the LEE. However, at angles of attack greater than the design value for which the PSSS was determined, the attachment of the LEE to the wing results in formation of a vortex system, emanating from sharp leading edges of the LEE device. The attendant low pressure, acting on the the LEE upper surface and the forward facing area of the wing leading edges, provides additional lift and reduced drag due to effective leading edge thrust recovery.

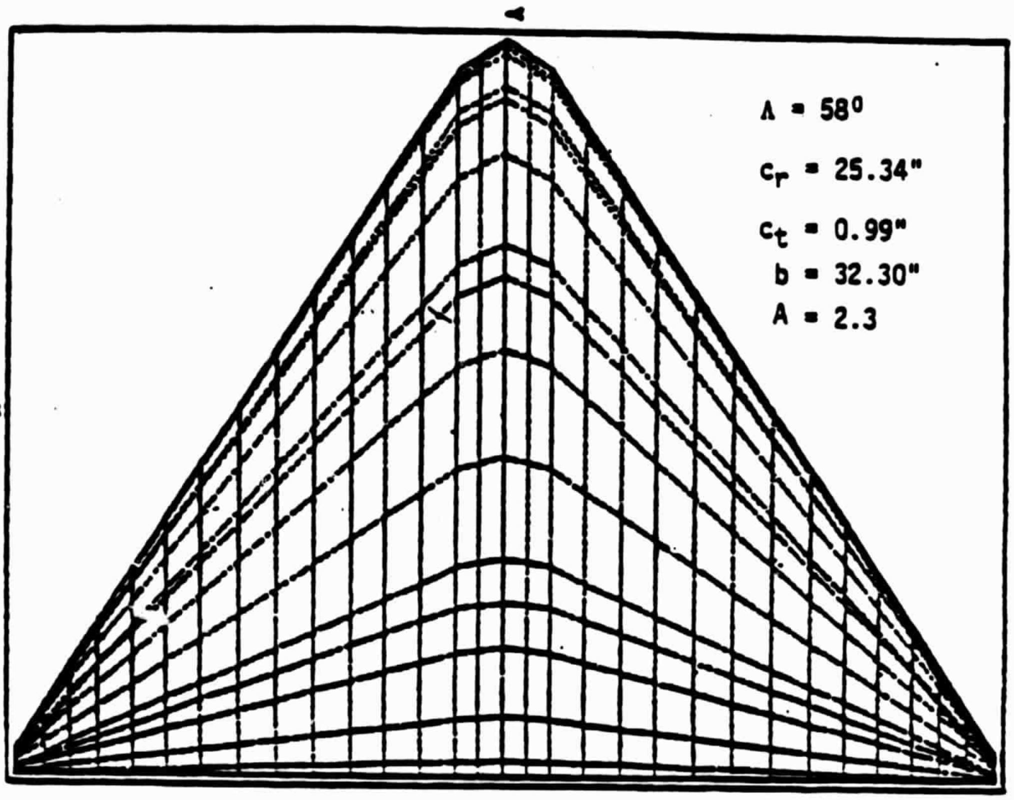

Figure 2. Planform of the wing model.

The second step in the LEE design involves the determination of the size and shape of an aerodynamically effective LEE for the given wing model. This is accomplished by employing the VLM-SA code for several LEE configurations with different constant chords, span extents and sweep angles. The aerodynamic performance of these various wing Lee configurations is used as the criterion for selecting an aerodynamically effective LEE.

The PAN AIR code was employed for the PSSS determination after earlier attempts to use the FVS and the VLM-SA codes proved unsuccessful (ref. 7). Sixteen spanwise stations were chosen, and a survey network was generated at each of these locations. The velocity components were determined at the central point of each panel of the survey network by executing the PAN AIR code. Figure 3 illustrates the typical survey network at station 4. The number of panels were augmented specially near the leading edge of the wing, by employing a computer code known as GEOMABS (ref. 13). This intensification of paneling is essential for accurately locating the pseudo stagnation streamline at any given spanwise station. From the array of the three velocity components thus obtained, the sidewash component (V_y) is neglected, and the resultant velocity vectors due to the axial (V_x) and the upwash (V_z) components are plotted at the central point of each panel of the given survey network. An example of this plot of $\sqrt{V^2_x + V^2_z}$ for the fourth station is shown in figure 4. The streamline associated with the minimum velocity magnitude ($|V_x| = |V_z| = 0$) is drawn tangent to the plotted velocity vectors. Since $V_y \neq 0$ at the point of minimum velocity magnitude, this point is referred to as the pseudo-stagnation point, and the streamline passing through this point is designated as the pseudo-stagnation

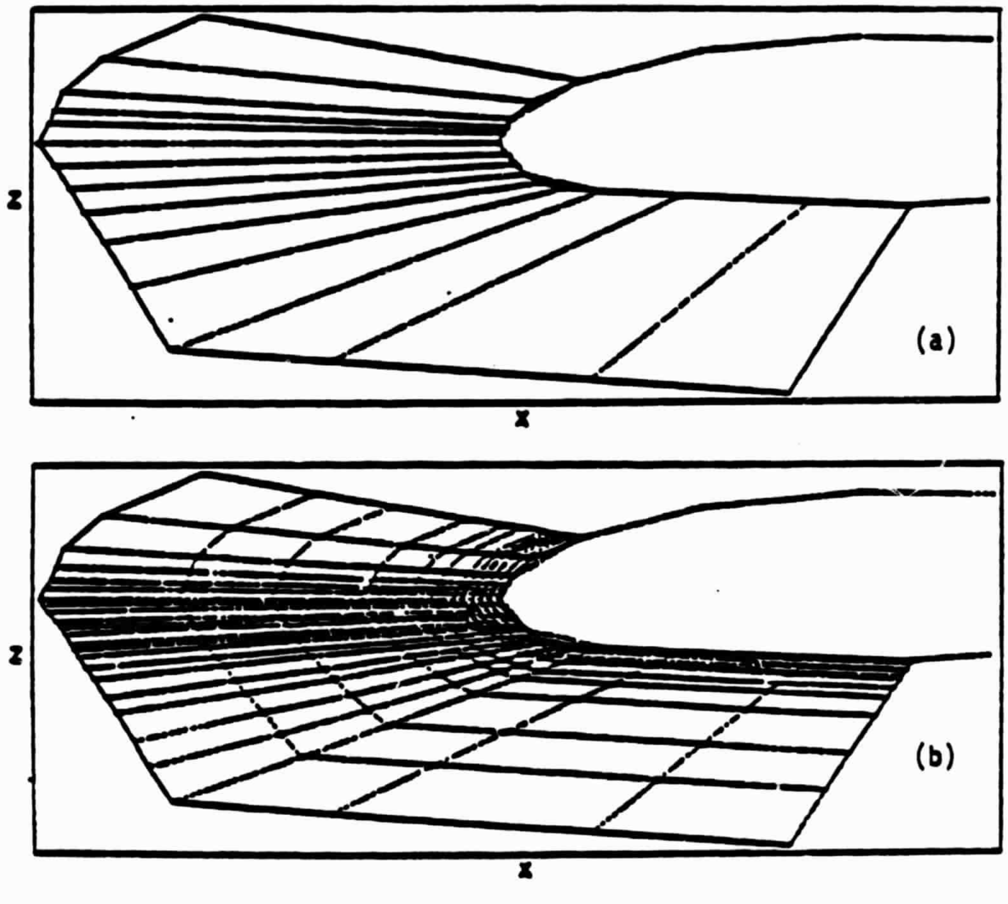

Figure 3. Enlarged view of survey network at station four
a) original panels,
b) densed panels.

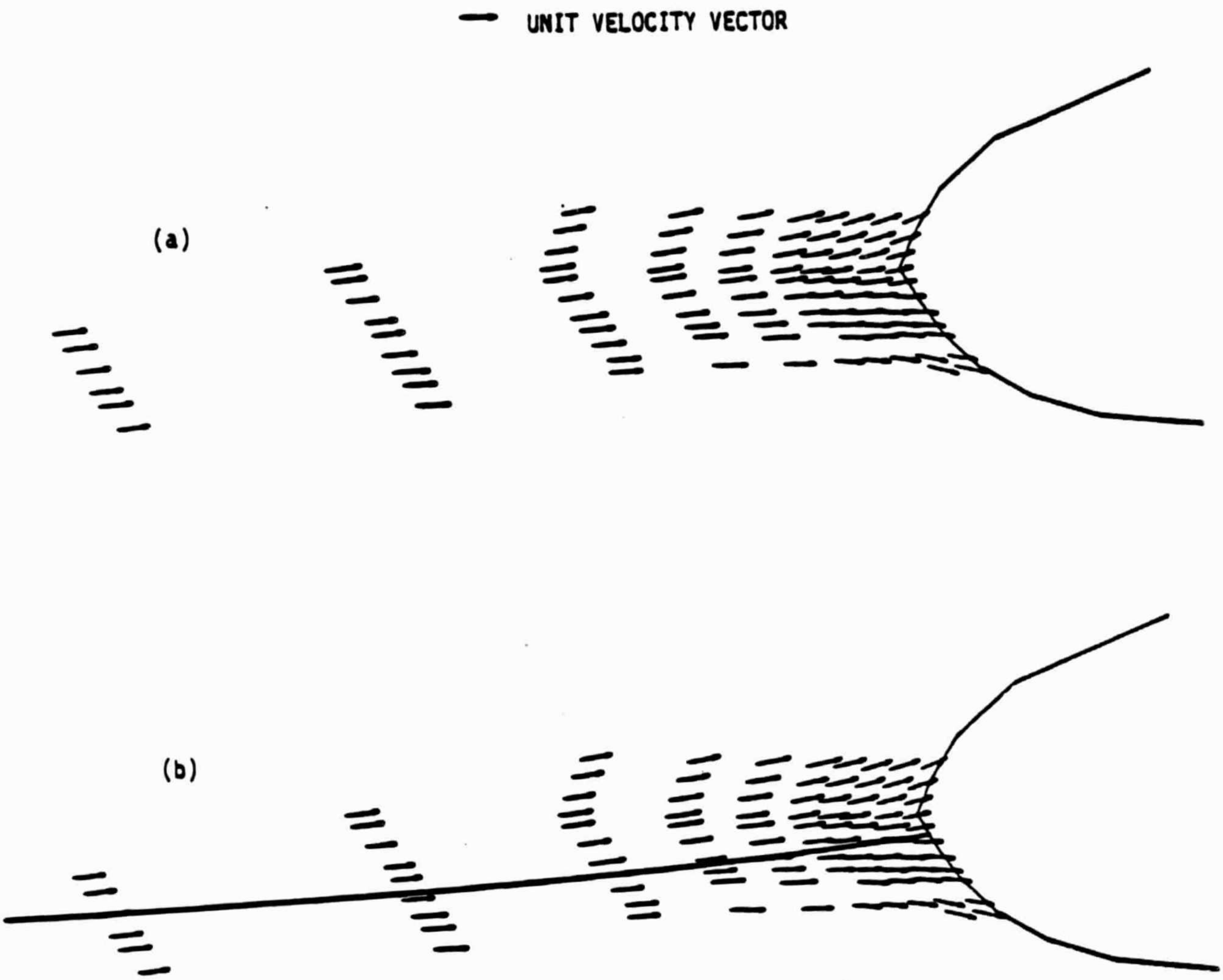

Figure 4. Fourth station,
a) velocity field solution,
b) pseudo-stagnation streamline solution.

streamline. This graphical procedure yields the coordinates of pseudo-stagnation streamline for the chosen spanwise station, and a cubic spline is fitted through the points to yield a smooth curve.

The above procedure is applied to other spanwise stations and for each case, the solutions extending-equally to a distance of 4.8" ahead of the wing leading edge were obtained. The pseudo-stagnation streamline solutions of various stations are joined by spanwise planar surfaces to yield the pseudo-stagnation stream surface solution. This is designated as the PSSS. The three views of the determined PSSS for the chosen wing at design condition are shown in fig. 5. It should be stated that unrealistic velocity field solutions were obtained at station 16 (corresponding to the wing tip) as a result of using zero wing thickness at the tip. Subsequently, the wing with the same planform and with non-zero thickness at the tip has been modeled and these results will be discussed later.

The accuracy of the determined PSSS solution was checked by employing the PAN AIR code for the wing-PSSS combination at design angle of attack, with the PSSS being specified as a lifting surface. The resulting ΔC_p values across the PSSS were generally very small, and this was taken to indicate that the determined PSSS solution is a good approximation of the actual dividing stream surface associated with the wing at the attached flow condition (ref. 7).

Once the PSSS is known, the size and shape of an aerodynamically effective LEE planform along the PSSS is determined by exercising the VLM-SA code for several wing LEE combinations. Prior to using the VLM-SA code, it was validated with regard to its capability of treating thick wings with leading edge extensions. The DM-1 model with and without the LEE was analyzed by the VLM-SA code (ref. 7) and the numerical results were compared with the data of Wilson and Lovell (ref. 12). For the DM-1, the results

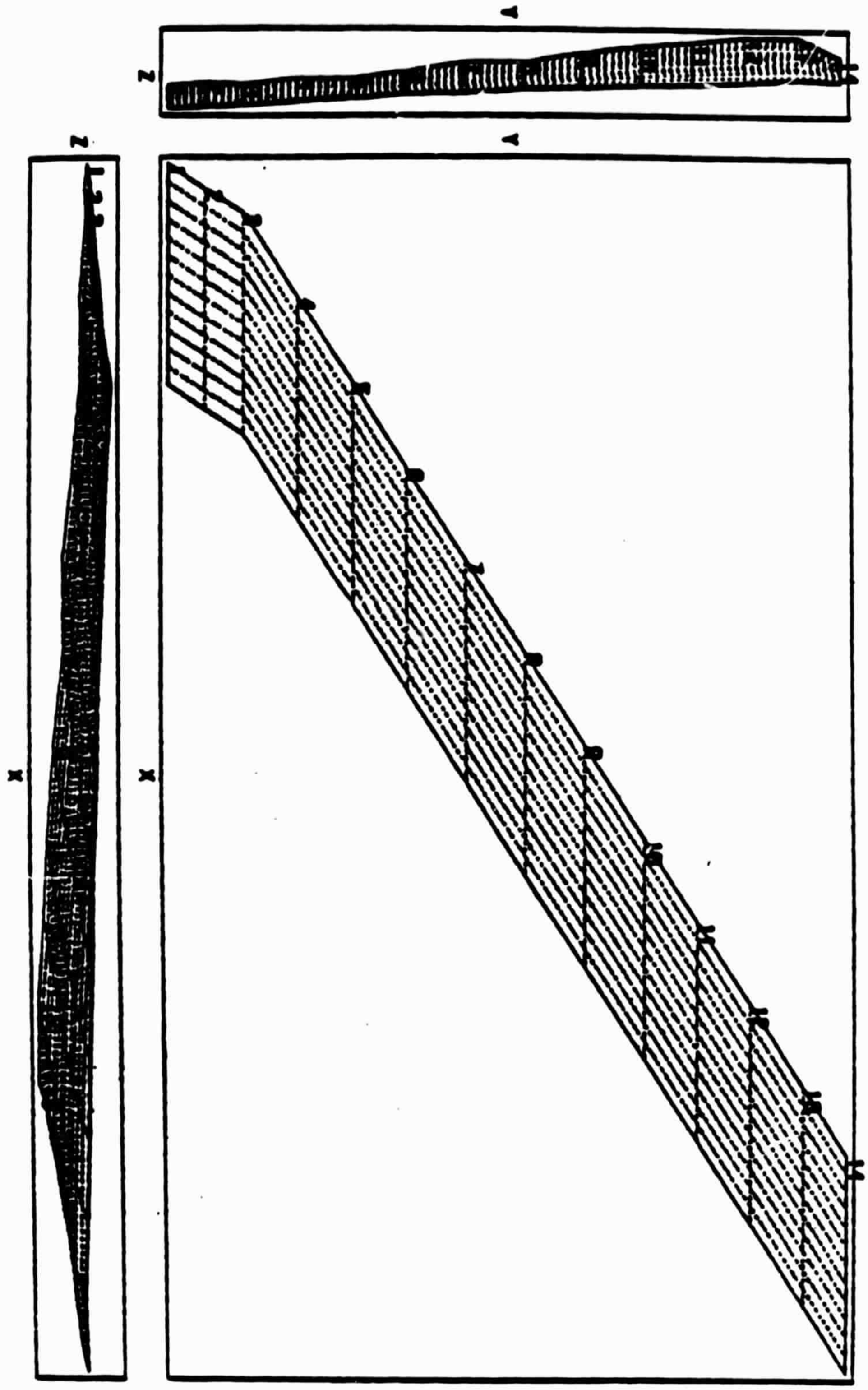

Figure 5. Three views of the determined PSSS solution.

predicted by the VLM-SA code were in good agreement with data up to a C_L value of 0.60. But for the case of DM-1 wing-LEE combination, the VLM-SA code overestimated the drag in the lift coefficient range of 0.5 to 0.8. (ref. 7). This difference is apparently due to the fact that the resulting VLM-SA solutions do not include the effect of the low pressure acting between the LEE and the upper surface maximum thickness line of the wing section to produce a thrust. Hence, the L/D ratio for the DM-1 wing-LEE combination is much lower than the data. Therefore, by analogy it is to be expected that the VLM-SA estimation of the drag would be higher in the wing-LEE analysis of the present study.

Six constant chord LEEs of chord dimensions 4.8", 3.6", 2.4", 1.6", 1.2" and 0.8" (Fig. 6), and six LEEs of sweep angles ranging from 53°, 54°, 55°, 56°, 57° and 58° (Fig. 7) were investigated for lift and drag coefficients. All selected LEEs initially had a semi-span of 14.33", which is also equal to 89.5% of the wing span. Subsequently, the VLM-SA estimates of lift and drag coefficients for 75% and 50% span LEEs with constant chords were also considered. The detailed results of these cases are given in ref. 7, and only a brief summary of results is given here.

1. Although increasing the LEE's constant chord reduces the total drag at moderate-to-high-lift coefficients, the L/D ratio showed little change with chord size.
2. At low and high lift coefficients, the variation of LEE sweep angle showed little effect on the drag. At moderate lift coefficients (i.e., 0.6 to 1.0), the LEEs with lower sweep angles were more effective in reducing the drag.

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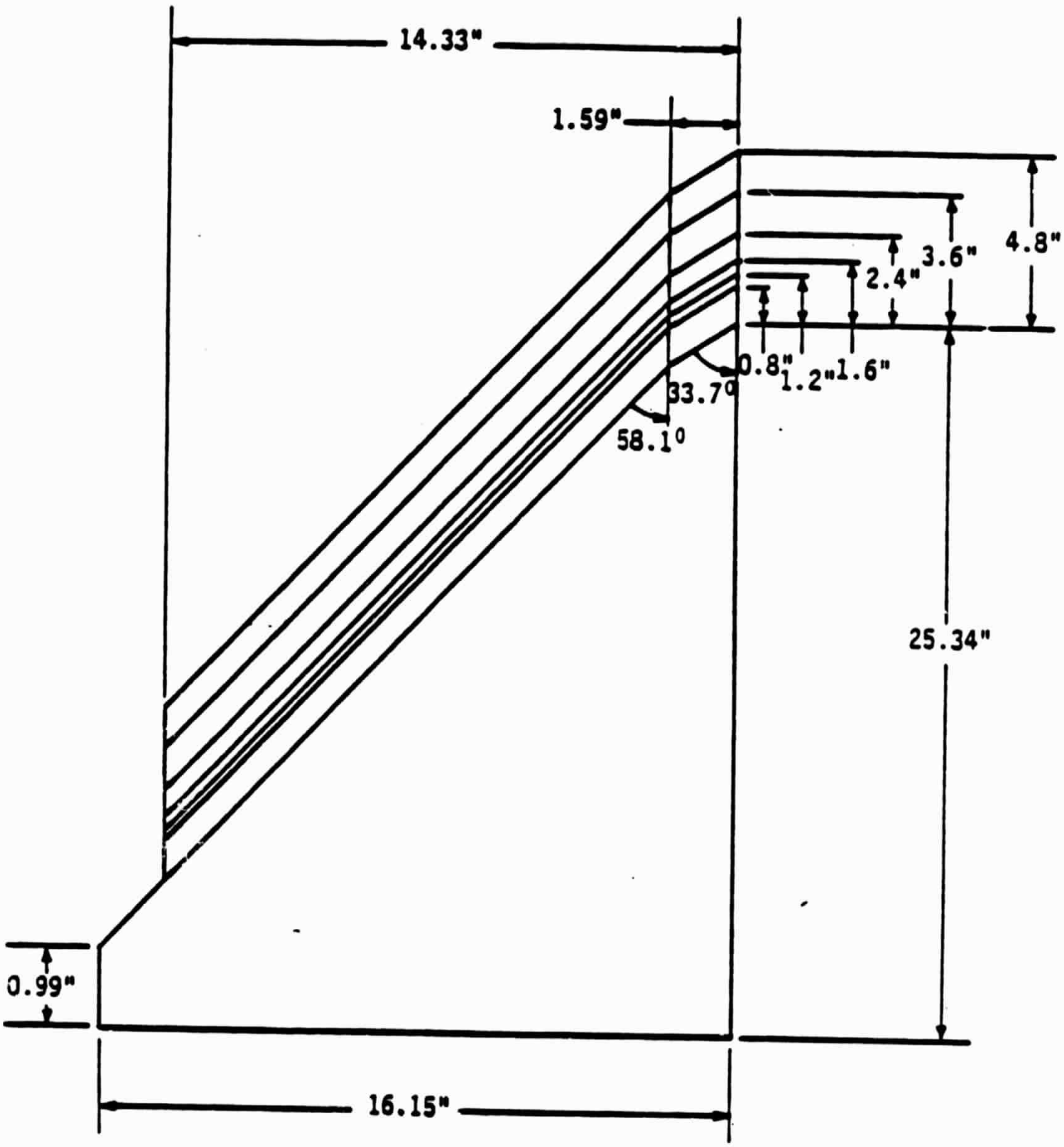

Figure 6. Schematic planform view of the wing model and the selected constant chord LEEs.

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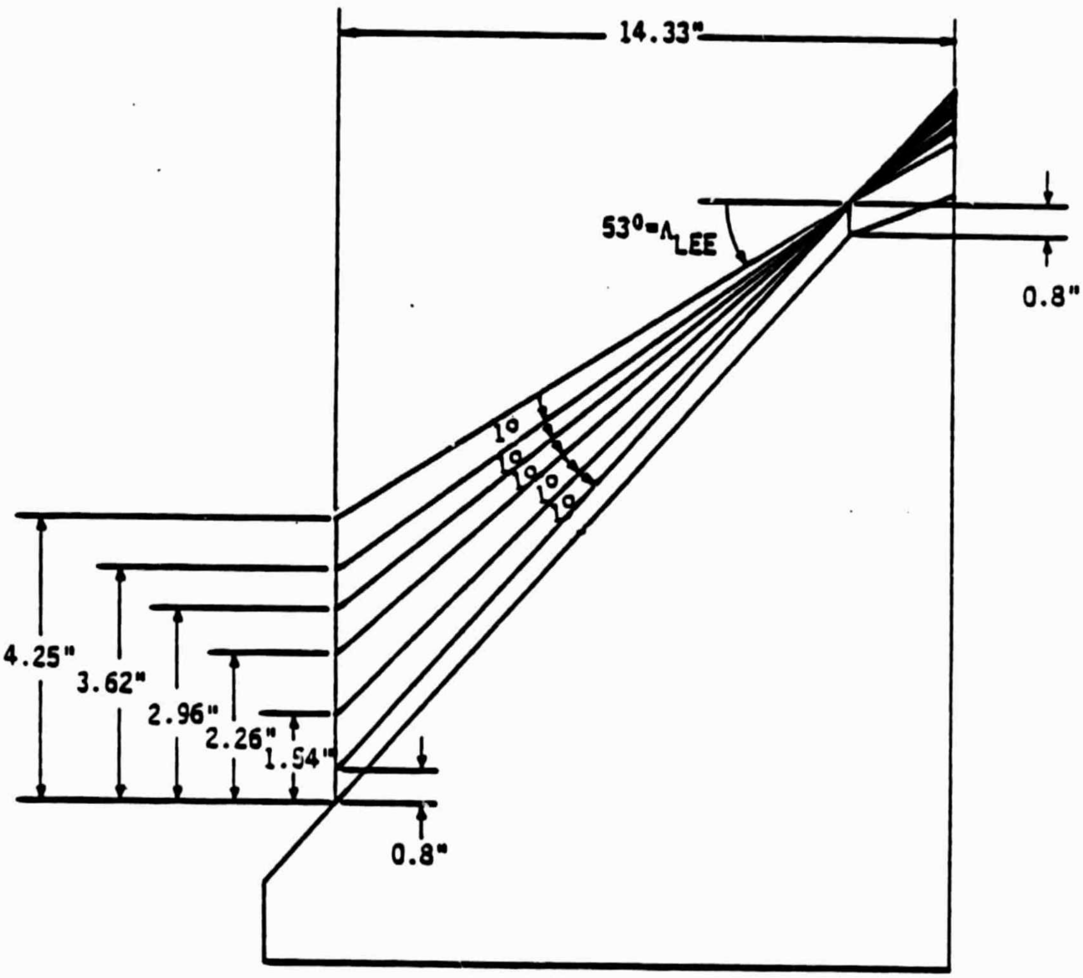

Figure 7. Schematic planform view of the wing model and the selected LEEs with different sweep angles.

3. The outboard reduction in the LEE span extent decreased the L/D ratio in the α range of 6.0 to 12°, regardless of the LEEs planform shape and area. However, this effect appeared to be insignificant beyond 14° angle of attack.

4. The design requirements of the present study, coupled with the results obtained for thirty six different LEE planforms suggest that 0.8" constant chord with 89% span extent has the aerodynamic potential of being selected as the candidate for the final LEE planform design. Further, relative to the above planform, the 1.2" constant chord, with same span extent, also appeared to be promising in achieving a better L/D ratio, especially at 10° angle of attack.

EFFECT OF FINITE WING TIP THICKNESS

As it was reported in ref. 7, the PAN AIR unrealistic flow field solution at the tip region of the thick delta wing prevented the graphical determination of the Pseudo-Stagnation Streamlines for the last two wing sections. As a result the determined Pseudo-Stagnation Stream Surface (PSSS) had a semi-span of 14.32", equal to 89% of the basic wing semi-span. In that reference it was also suggested that perhaps this problem was due to the choice of zero wing thickness at the tip section. This problem was re-examined by adding a 0.01" thickness to the tip section of the wing model. The PAN AIR computer code was employed to determine the effect of the additional tip-thickness on the pressure distribution over the wing surface as well as on the net lifting-pressure across the graphically determined PSSS. This solution indicated that only a slight change in pressure occurs around the leading edge portion of the outboard sections (beyond 10th station) of the wing model. Although the pressures around the tip region of the wing model are still unrealistic, it is important to mention that they are improved compared to the case with zero tip-thickness. This solution also showed almost no change in the ΔC_p across the PSSS which was determined earlier with zero thickness at the tip section (ref. 7).

Attempts were also made to extend the PSSS from 89% span extent to the wing full span with finite wing thickness at the tip-section. As a result, a survey network was generated for a station slightly inboard (y = 16.10") of the wing tip-section and the PAN AIR code was employed to determine the velocity field solutions for the section at the design condition (α = 6.0°, M = 0.8). The plotted velocity vectors indicated that the pseudo-stagnation point (i.e, $|V_x| \approx |V_z| \approx 0$) occurs on the upper surface

of the wing section, at a distance of about half of the local chord from the leading edge. This effect created PAN AIR modeling (edge abutments) problems, because the pseudo-stagnation point moves from the wing lower surface at sections away from the wing tip to upper surface near the tip section. As a result, the determined PSSS can not be extended all the way to the tip section.

CONCLUSIONS

The three vortex codes considered in this study, namely VLM-SA, FVS and QVL, were reasonably successful in predicting the aerodynamic performance of flat wings at low to moderate angles of attack. The pitching moment was generally overpredicted by all three codes. For highly cambered wings, these codes had only limited success in yielding converged solutions. The FVS method was least successful in providing converged solutions for highly cambered configurations. Although QVL and VLM-SA methods were more successful in providing converged solutions, the numerical results were not always in good accord with the available data. For the two cases that included wing thickness, in one case - the SCAT-15 wing model, the solution obtained from the FVS method was not in good accord with the data. In the second case, e.g., the thick DM-1 wing-LEE combination, the solution convergence was not achieved. From the limited experience gained from modeling the wing thickness, it appears that more studies are required before any firm conclusions can be drawn regarding the code capabilities. The VLM-SA code generally predicted higher values of C_L and C_D in the higher angle of attack range when compared to QVL and FVS methods. However, the lack of experimental results in the higher angle of attack range precluded any firm conclusions regarding the code capabilities even for the flat wing.

The LEE analytical design methodology presented in this study, has been applied to size an aerodynamically effective LEE for reducing the drag on a thick delta wing. The LEEs with constant chords of 0.8" and 1.2" dimensions appear to be best suited for the wing selected in this study. The LEE design procedure could not be validated due to lack of data for wing-LEE

combination. It is recommended that a wind-tunnel study be carried out for the chosen wing-LEE combination to put the LEE analytical design procedure on a firm footing. Future efforts in this area should also address the question of development of appropriate analytical tool(s) for accurate determination of the PSSS at a given design flow-condition, as well as, the LEE planform optimization.

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9. Reddy, C.S.: Investigation of Aerodynamic Characteristics of Wings Having Vortex Flow Using Different Numerical Codes. NASA CR-165706, 1981.

10. Reddy, C.S.: Aerodynamic Performance of Slender Wings with Separated Flows. Technical Report, Old Dominion University Research Foundation, Norfolk, VA, 1982.

11. Reddy, C.S.; and Ghaffari, F.: Study of Highly Sweptback Wings by the Free Vortex Sheet Method. Technical Report, Old Dominion University Research Foundation, NASA Research Grant NSG 1561, 1982.

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13. Hall, J.F.; Neuhart, D.H.; and Walkley, K.B.: An Interactive Program for Manipulation and Display of Panel Method Geometry. NASA CR-166098, March 1983.

APPENDIX A

Complete List of Grant Related Publications During the Period 1978-84

1. Reddy, C.S: Theoretical Study of Aerodynamic Characteristics of Wings Having Vortex Flow. NASA CR-159184, 1979.

2. Reddy, C.S.: Investigation of Aerodynamic Characteristics of Wings Having Vortex Flow Using Different Numerical Codes. NASA CR-165706, 1981.

3. Reddy, C.S.: Aerodynamic Performance of Slender Wings Having Vortex Flow. Technical Report, Old Dominion Research Foundation, Norfolk, Va., 1982.

4. Reddy, C.S.; and Ghaffari, F.: Study of Highly Sweptback Wings by the Free Vortex Sheet Method. Technical Report, Old Dominion University Research Foundation, 1982.

5. Reddy, C.S.: Effects of Leading-Edge Vortex Flaps on Aerodynamic Performance of Delta Wings. J. of Aircraft, vol. 18, No. 9, 1981.

6. Reddy, C.S.: Analytical Investigation of Aerodynamic Characteristics of Highly Swept Wings with Separated Flow. J. of Aircraft, vol. 17, No. 1, 1980.

7. Ghaffari, F; and Chaturvedi, S.K. An Analytical Design Procedure for the Determination of Effective Leading Edge Extensions on Thick Delta Wings. Final Report for NASA Research Contract NAS-1-17099, Task Authorization No. 26, 1984. Also M.S. Thesis of F. Ghaffari with same title, submitted to Old Dominion University for partial fulfilment of M.S. degree requirement.

8. The present report - As Final Report for NASA NSG-1561 Grant.